NASA/TM—2002-211555

Status of the RBCC Direct-Connect Mixer Combustor Experiment

James F. Walker, Hani Kamhawi, Thomas M. Krivanek,
Scott R. Thomas, and Timothy D. Smith
Glenn Research Center, Cleveland, Ohio

May 2002

The NASA STI Program Office . . . in Profile

Since its founding, NASA has been dedicated to the advancement of aeronautics and space science. The NASA Scientific and Technical Information (STI) Program Office plays a key part in helping NASA maintain this important role.

The NASA STI Program Office is operated by Langley Research Center, the Lead Center for NASA's scientific and technical information. The NASA STI Program Office provides access to the NASA STI Database, the largest collection of aeronautical and space science STI in the world. The Program Office is also NASA's institutional mechanism for disseminating the results of its research and development activities. These results are published by NASA in the NASA STI Report Series, which includes the following report types:

- TECHNICAL PUBLICATION. Reports of completed research or a major significant phase of research that present the results of NASA programs and include extensive data or theoretical analysis. Includes compilations of significant scientific and technical data and information deemed to be of continuing reference value. NASA's counterpart of peer-reviewed formal professional papers but has less stringent limitations on manuscript length and extent of graphic presentations.
- TECHNICAL MEMORANDUM. Scientific and technical findings that are preliminary or of specialized interest, e.g., quick release reports, working papers, and bibliographies that contain minimal annotation. Does not contain extensive analysis.
- CONTRACTOR REPORT. Scientific and technical findings by NASA-sponsored contractors and grantees.
- CONFERENCE PUBLICATION. Collected papers from scientific and technical conferences, symposia, seminars, or other meetings sponsored or cosponsored by NASA.
- SPECIAL PUBLICATION. Scientific, technical, or historical information from NASA programs, projects, and missions, often concerned with subjects having substantial public interest.
- TECHNICAL TRANSLATION. English-language translations of foreign scientific and technical material pertinent to NASA's mission.

Specialized services that complement the STI Program Office's diverse offerings include creating custom thesauri, building customized data bases, organizing and publishing research results . . . even providing videos.

For more information about the NASA STI Program Office, see the following:

- Access the NASA STI Program Home Page at ***http://www.sti.nasa.gov***
- E-mail your question via the Internet to help@sti.nasa.gov
- Fax your question to the NASA Access Help Desk at 301–621–0134
- Telephone the NASA Access Help Desk at 301–621–0390
- Write to:
  NASA Access Help Desk
  NASA Center for AeroSpace Information
  7121 Standard Drive
  Hanover, MD 21076

NASA/TM—2002-211555

Status of the RBCC Direct-Connect Mixer Combustor Experiment

James F. Walker, Hani Kamhawi, Thomas M. Krivanek,
Scott R. Thomas, and Timothy D. Smith
Glenn Research Center, Cleveland, Ohio

Prepared for the
Combustion, Airbreathing Propulsion, Propulsion Systems Hazards,
and Modelling and Simulation Subcommittees Joint Meeting
sponsored by the Joint Army-Navy-NASA-Air Force
Destin, Florida, April 8–12, 2002

National Aeronautics and
Space Administration

Glenn Research Center

May 2002

The Aerospace Propulsion and Power Program at
NASA Glenn Research Center sponsored this work.

Available from

NASA Center for Aerospace Information
7121 Standard Drive
Hanover, MD 21076

National Technical Information Service
5285 Port Royal Road
Springfield, VA 22100

Available electronically at http://gltrs.grc.nasa.gov/GLTRS

Status of the RBCC Direct-Connect Mixer Combustor Experiment

James F. Walker, Hani Kamhawi, Thomas M. Krivanek,
Scott R. Thomas, and Timothy D. Smith
National Aeronautics and Space Administration
Glenn Research Center
Cleveland, Ohio 44135

ABSTRACT

The NASA Glenn Research Center is developing hydrogen based combined cycle propulsion technology for a single-stage-to-orbit launch vehicle application under a project called GTX. Rocket Based Combined Cycle (RBCC) propulsion systems incorporate one or more rocket engines into an airbreathing flow path to increase specific impulse as compared to an all rocket-powered vehicle. In support of this effort, an RBCC direct-connect test capability was established at the Engine Components Research Laboratory to investigate low speed, ejector ramjet, and initial ramjet operations and performance. The facility and test article enables the evaluation of two candidate low speed operating schemes; the simultaneous mixing and combustion (SMC) and independent ramjet stream (IRS). The SMC operating scheme is based on the fuel rich operations of the rocket where performance depends upon mixing between the rocket plume and airstream. In contrast, the IRS scheme fuels the airstream separately and uses the rocket plume to ignite the fuel-air mixture. This paper describes the test hardware and facility upgrades installed to support the RBCC tests. It also defines and discusses low speed technical challenges being addressed by the experiments. Finally, preliminary test results, including rocket risk mitigating tests, unfueled airflow tests, and the integrated system hot fire test will be presented.

INTRODUCTION

The NASA Glenn Research Center is developing technology for a hydrogen-fueled rocket based combined cycle (RBCC) propulsion system because it may enable single-stage-to-orbit (SSTO) missions and realize substantially reduced launch costs.[1] The combined cycle propulsion system, designed to operate in multiple modes, must collectively produce a certain level of performance to achieve the overall system and mission requirements. Low speed, mode 1, operations are particularly challenging because it is the only mode where the rocket element and air-breathing flow path operate simultaneously. In addition, propulsion demands are high because the vehicle requires maximum thrust while experiencing its lowest fuel efficiency. With the use of hydrogen, the importance of low speed fuel efficiency is heightened due to fuel tank volume considerations. Consequently, small improvements in low speed performance will have significant system benefits. This is a formidable challenge because the flow path operating characteristics vary considerably from lift off to ramjet transition. To address these challenges, several different cycle concepts have been proposed. Two approaches reported in the literature are diffusing and afterburning (DAB) and simultaneous mixing and combustion (SMC).[2] Another one conceived for the GTX flowpath is called the independent ramjet stream (IRS).[3] To investigate the different cycle concepts and determine low speed operations and performance, a direct-connect mixer combustor experiment was defined. This paper discusses the technical challenges of low speed RBCC operations and presents the status of the experimental effort.

LOW SPEED TECHNICAL CHALLENGES

The cycle with the highest potential thermodynamic efficiency is the DAB scheme.[2] It achieves high performance because it separates the mixing and combustion processes. However, the DAB cycle requires a more complicated flowpath than alternate approaches. Furthermore, it is unattractive for hypersonic applications because it requires a physical throat located downstream of the mixing section, which necessitates variable geometry for scramjet operations. In contrast, the SMC cycle can accommodate ramjet and scramjet operations without the need for a physical throat. Instead, it utilizes a thermal throat generated by the mixing and burning process. In this approach, the rocket plume and inlet airstream are designed to mix and burn simultaneously. Thus, it is one operating scheme being investigated for the GTX concept. Figure 1 depicts the GTX propulsion concept operating with SMC.

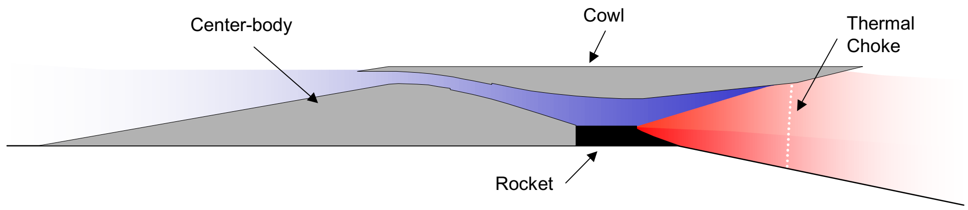


Figure 1 - Simultaneous Mixing and Combustion.

The other low-speed operating scheme being considered is the IRS cycle.[3] IRS was defined to simplify the RBCC mechanical and operational design. Specifically, the SMC requires the rocket and airstream to completely mix, which necessitates multiple rocket elements and/or a longer mixer duct. In contrast, the GTX concept uses a single rocket element in each engine module with fuel injected directly into the airstream through the ramjet and scramjet wall injectors once supersonic speeds are reached. Numerical studies reported in references 3 and 4, indicate no benefit to adding fuel directly to the airstream at subsonic speeds. The rocket, which is operated close to stoichiometric mixture ratio to optimize engine performance and reduce the vehicle's propellant tank volume, is used to ignite the fuel-air mixture. This scheme, which is depicted in figure 2, can achieve good performance without complete mixing of the two streams based on computational analysis reported in reference 4. The analysis is mainly focused on flame front propagation, stability, and propulsive performance. The experimental effort will investigate the technical challenges with practical injection schemes and validate the numerical models.

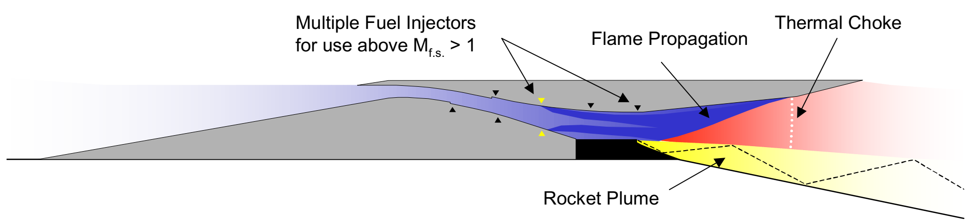


Figure 2 - Independent Ramjet Stream.

RBCC operating characteristics are particularly challenging at lift off and subsonic flight speeds because the ram/scramjet duct shrouds the rocket so ejector action can cause over-expansion losses. To prevent thrust degradation, the inlet must provide sufficient airflow to ventilate the rocket plume. With the GTX concept, this is accomplished by moving the center-body forward to increase flow area and minimize flow restriction. As flight speeds surpass sonic conditions, ram pressure rises so fuel can be injected directly into the airstream and burned to generate ramjet thrust. This thrust augmentation increases rapidly with Mach number, enabling higher specific impulse performance by throttling the rocket. Managing the proportion of thrust generated by the rocket and ramjet permits optimization of propulsion performance and vehicle thrust requirements. Throughout low speed operations, it is important to control the inlet diffuser exit pressure because it governs inlet performance. This is accomplished by controlling the location of the thermal throat by adjusting the amount and distribution of fuel injected directly into the airstream. Preliminary analysis reported in reference 4 substantiates this approach, but it still remains to be demonstrated with an experimental effort.

At flight speeds between Mach 2.5 to 3.0, the rocket can be completely shut off because ramjet thrust is sufficient to accelerate the vehicle with improved specific impulse. The transition to ramjet operations is another technical challenge to investigate with the direct-connect experiments. Specifically, maintaining the flame during the mode transition will present a challenge. The GTX concept is designed to use an aft facing flat surface adjacent to the rocket nozzle as the flame holder. As a contingency, fuel struts can be used to hold the flame front, but present added complexity for high-speed ramjet and scramjet operations. Obviously, they would need to be retractable or expendable. Otherwise, they would cause undesirable flow drag and cooling issues at higher flight speeds.

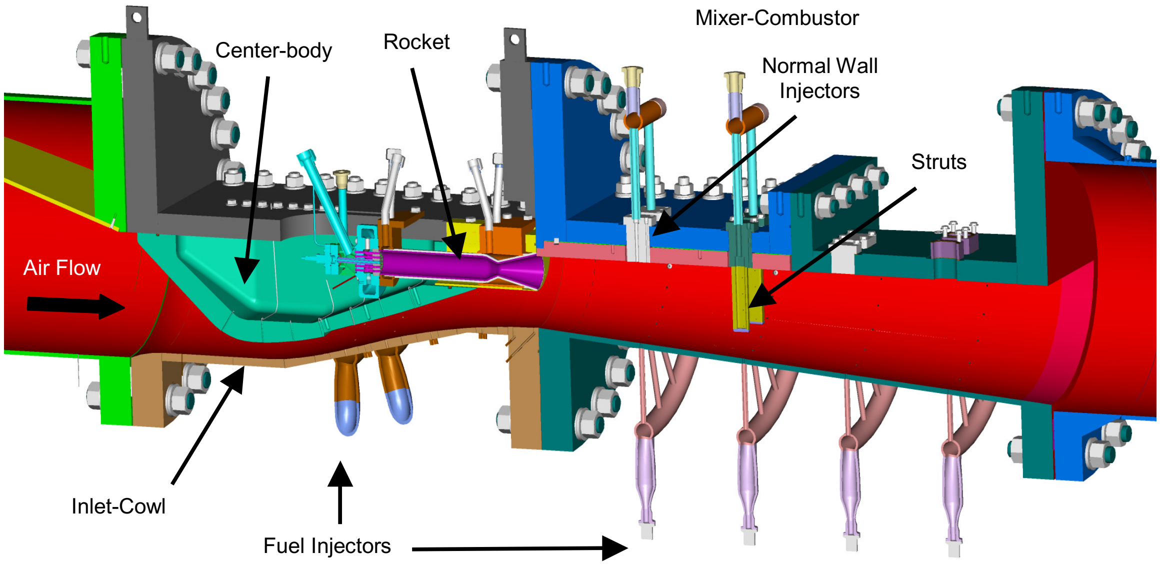


Figure 3 - Direct-Connect Mixer-Combustor Rig.

TEST ARTICLE

A direct-connect mixer combustor was designed and built to parametrically investigate low speed operations and performance. The test article consists of a mixer-combustor (combustor) section, an inlet-cowl section, a moveable center-body, and a water-cooled rocket engine as displayed in figure 3.

Three combustors measuring 25-in., 50-in., and 100-in. long, were fabricated to evaluate mixing and engine performance for varying duct lengths. The combustor sections, located aft of the rocket nozzle, use a half conical cowl geometry closed out with a flat plate on top as shown in figure 3. The model forms a half circular flowpath, and the cross sectional area for all three combustors increases from 69.9 in.^{2} to 191.1 in.^{2} This results in a different expansion angle for each combustor; 2.5° for the 100-in. combustor, 5.0° for the 50-in. combustor, and 9.9° for the 25-in. combustor. The full GTX nozzle expansion was not duplicated to permit a larger scale model in the test facility. Instead, it was truncated at the capture area value since the experiment focuses on mixing and combustion. To investigate IRS and ramjet performance, the combustor sections contain wall injection stations spaced 10 in. apart down their length with the exception of the 100-in. combustor, which increases the spacing to 20 in. for the 40, 60, and 80-in. locations. At each station, there are eleven, 0.2-in. diameter holes equally spaced around the cowl's circumference. At each axial station, a common manifold that is offset from the combustor wall supplies the injectors. In addition, the flat plate side of the combustor can accommodate normal injectors or strut injectors. Figure 3 shows normal injectors at the first fuel station and struts installed in the second one. The normal injectors contain three, 0.332-in. diameter holes located across the flow path. The strut injectors consist of three equally spaced struts. Each strut has four, 0.125-in. diameter holes on each side to inject hydrogen normal to the airflow. The struts also use a flat back face to provide flame-holding capability. The combustors have nominally eighty-three wall static pressure taps to measure centerline and off-centerline pressures. They also contain about seventeen embedded wall thermocouples to measure hot sidewall temperatures.

Three vertical rakes are mounted at the end of the combustor for pressure measurements. Each rake has seven probes. The center rake uses larger-sized ports to support the gas sampling system. Specifically, the rake is connected to a bottle storage system to collect gas samples at prescribed times during a test. Analysis of the collected samples will provide information on the distribution of mixing between the fuel rich rocket stream and airstream. This will also provide insight into the mixing pattern between the airstream and the hydrogen fuel injected directly into the airstream. The bottle storage system contains four sets of seven sample bottles, which are connected to the seven probes on the center rake. This allows the collection of four separate gas sample sets. The sample collection system is operated remotely, and it allows remote sample bottle evacuation if it is decided to replace a collected sample set. Analysis of the gas samples will be performed using a standard gas chromatograph (GC) with a thermal conductivity detector (TCD). The GC/TCD is configured to determine the mole fraction of H_2, N_2, and O_2 in the gas sample. It was calibrated using known samples containing different mole fractions of H_2-N_2 and N_2-O_2 mixtures at different pressures. Figure 4 shows a schematic of the gas collection and analysis setup.

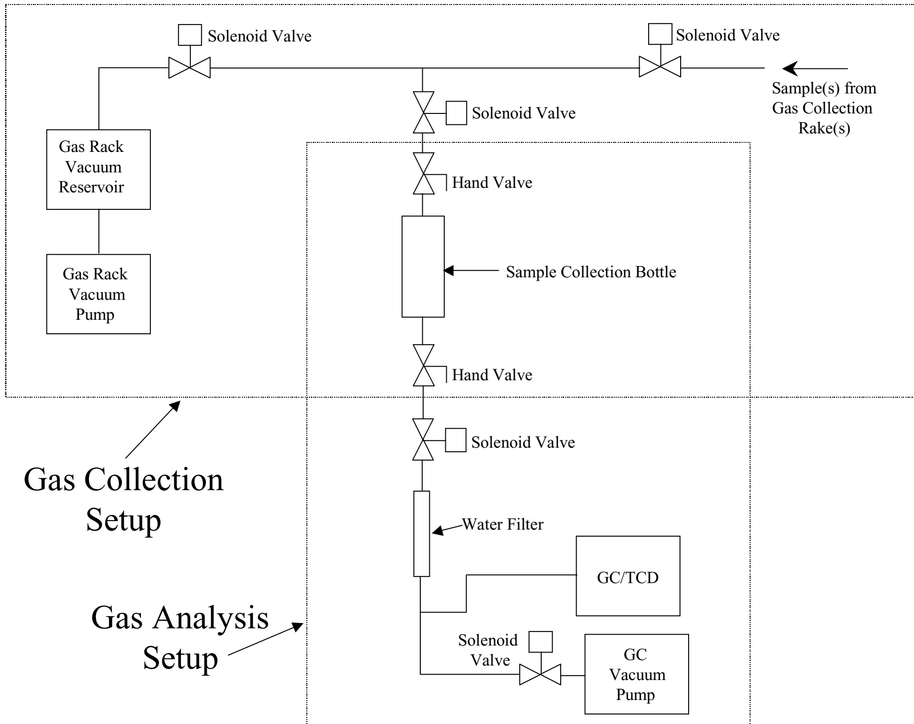


Figure 4 - Representative schematic of a single gas sample collection and analysis setup.

The inlet-cowl section, which is the section forward of the combustor flange, contains two additional normal fuel injection stations. Each contains eleven, 0.2-in. diameter holes equally spaced. This design permits premixing the fuel and air upstream of the mixer-combustor as conceptualized for the IRS cycle. The cowl has nominally twenty-nine wall static pressure taps measuring centerline and off-centerline pressures, and it contains three embedded wall thermocouples.

The center-body can be repositioned to simulate the flowpath configuration corresponding to different points along the flight trajectory. This is accomplished by loosening several bolts attaching the center-body to the model and sliding it to the desired position. Thus, the position is fixed during a test series. The center-body contains a backward facing step measuring 0.333-in. high used to provide isolation between the inlet and combustor. Aft of the step, the flow contour is representative of the GTX configuration. Forward of the step location, the center-body contour was designed to smoothly transition the flow from pipe flow to the half annular flow at the step. This is accomplished by first transitioning pipe flow to a half pipe flow. Then, the front face of the center-body uses a half 45° cone to transition the flow to the annular flow. The center-body contains eight wall static pressures and three thermocouples.

The center-body slides along a fixed hub section, which contains the rocket element. The rocket hardware consists of an injector, a torch igniter, a water-cooled nozzle, and a water-cooled combustion chamber. Due to flow blockage and base drag concerns, the amount of space allotted for rocket placement was limited as shown in figure 3. As a result, inlet propellant lines and feed locations were uniquely designed to fit into the required space. In addition, the inlet and exit water-cooling lines to the rocket were routed from one side of the engine through two manifolds due to space considerations. Three lines were used for both inlet and exit to provide uniform coolant flow. A photograph of the actual rocket installation is displayed in figure 5.

The injector, which is shown in figure 6, has a 2.055-in. diameter, porous sintered wire mesh faceplate for hydrogen flow and twenty-four 0.093-in. diameter oxygen injection elements. Ignition is initiated with a gaseous hydrogen/gaseous oxygen torch ignitor.[5] The ignitor tube exits through the center of the injector as display in figure 6. The chamber pressure is measured at the injector face with a 0.125-in. inner diameter tube that is brazed into the hydrogen manifold.

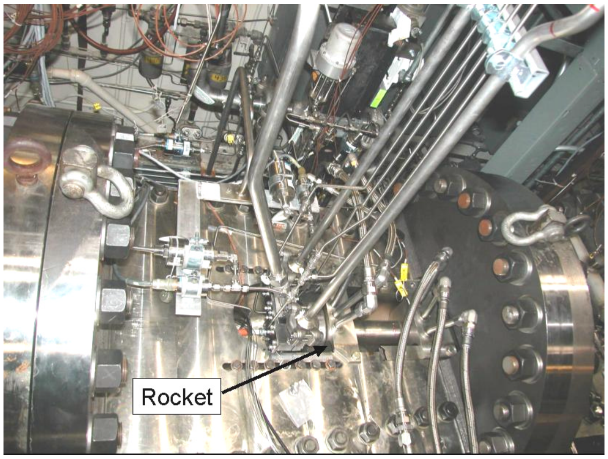


Figure 5 - Rocket mounted in Center-body.

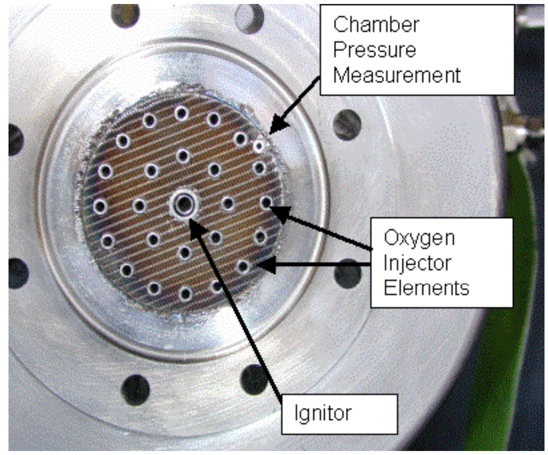


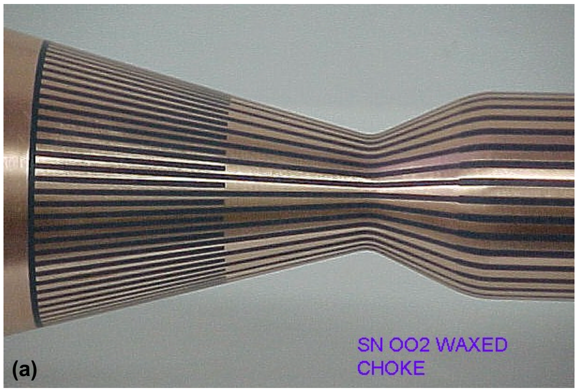
Figure 6 - Rocket engine injector.

Parameter	Dimension
Throat diameter (in.) =	1.0
Chamber diameter (in) =	2.055
Contraction ratio =	4.223
Exit diameter (in) =	3.0
Exit area Ratio =	9.0
Exit angle (degrees) =	15.0
Total chamber length (in) =	15.0
Chamber barrel section length (in) =	10.0
Number of injection elements =	24
Injection element diameter (in) =	0.093
Number of water cooling passages =	36

Table 1 - Rocket combustion chamber dimensions.

The rocket design utilized a single assembly combustion chamber. The water-cooled chamber is 15-in. long with a 1.0-in. throat diameter, and an internal diameter of 2.055 in. The internal combustion chamber liner was made from oxygen free copper with an electroformed nickel structural jacket. The expansion surface is a 15° half-angle conical nozzle with a total area ratio of nine. Table 1 summarizes the hardware dimensions.

The combustion chamber is cooled by 36 pass-and-a-half milled water passages. Due to the unique configuration of the direct-connect test article, the water inlet manifold was required to be upstream of the nozzle exit location. Water entered the cooling jacket 2.0-in. upstream of the nozzle exit and flowed parallel to the combustion gases. Flow was then turned 180° at the nozzle exit and flowed counter to the combustion gases, exiting the chamber just upstream of the injector face location. The coolant channel cross-section was 0.070x0.103-in. from the inlet to the beginning of the convergent section of the nozzle. For the barrel section to the exit, the coolant channels cross-section was 0.093x0.103-in. Figures 7(a) and (b) show the machined combustion chamber coolant passages. Hot-gas wall thickness was 0.050-in. for the entire chamber.



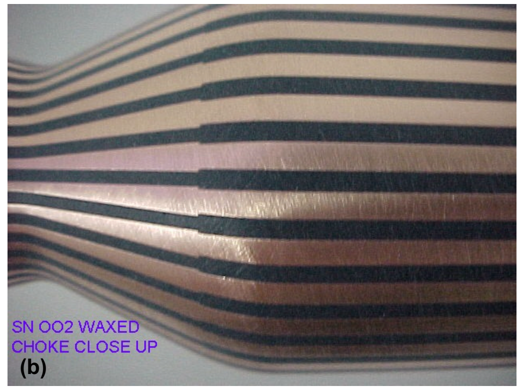


Figure 7 - Machined combustion chamber water cooling passages.

TEST FACILITY

The Engine Component Research Laboratory (ECRL) was selected for the direct connect tests because of its capabilities.[6] Specifically, it is capable of providing non-vitiated air heated to temperatures up to 600 °F and simulating up to 50,000 ft altitude by utilizing the NASA Glenn Research Center's altitude exhaust capabilities. This permits ECRL to simulate ejector ramjet and ramjet mode flight conditions up to Mach 3. However, the rocket operations required modifications to the existing hydrogen system and required the installation of a new oxygen system and high-pressure water-cooling system. Figure 8 displays the test cell in ECRL containing the direct-connect experiment.

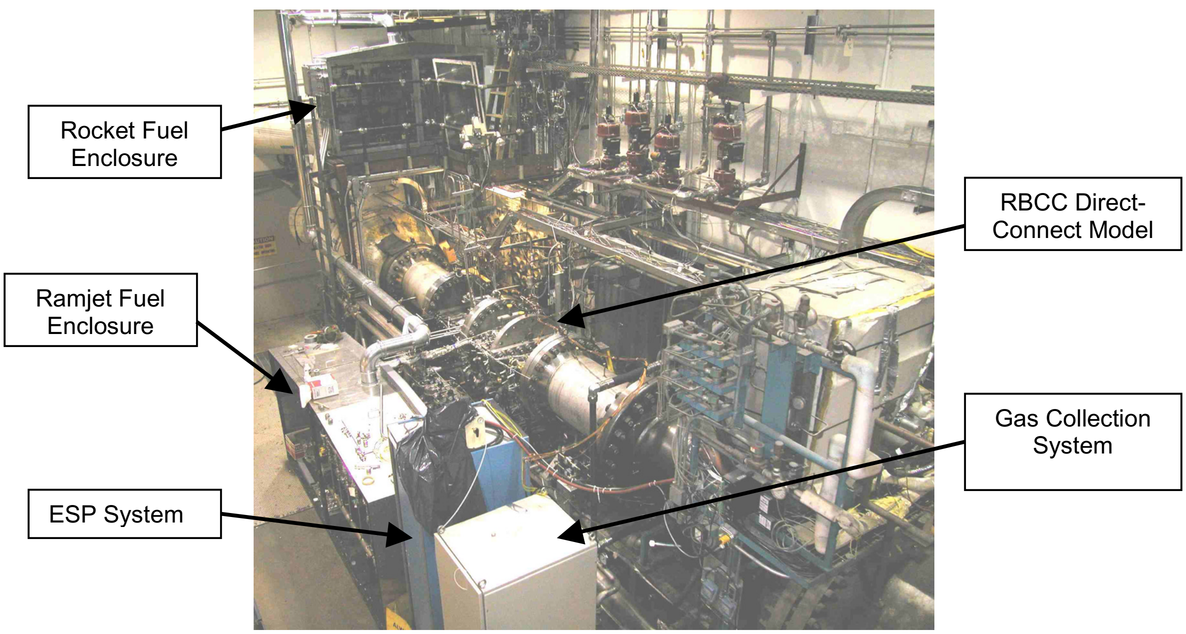


Figure 8 - Test Facility with RBCC model installed.

A hydrogen system that was recently installed needed modifications to support the higher supply pressures required by the rocket. In addition, a separate supply line for a metered fuel injection system was designed and built to supply hydrogen to the multiple fuel injection ports located on the model. The system was designed to provide up to 1.0 lbm/sec flow rate at full trailer pressure of 2400 psi. Outside the building, the flow is metered to prevent higher flow rates for safety reasons. In addition, components and line connections that can potentially leak were located in sealed enclosures. There are two enclosures located inside the cell as displayed in figure 8. One contains the regulators and valves used for the rocket fuel system, and the other denoted the ramjet fuel enclosure contains the regulators and valves used to inject hydrogen directly into the airstream. The ramjet enclosure contains multiple sets of paired sonic venturi meters and valves to supply individual model fuel injection stations, permitting fuel flow rate step increases during a single test. These enclosures along with two additional ones located on the roof contain

vents that exhaust to safe locations outside the test cell. Hydrogen detectors located in the vents automatically terminate the test if a hydrogen leak occurs. Two portable racks located next to the ramjet enclosure contain the electronically scanned pressure system (ESP) and gas sample collection system. The ESP system was configured to collect up to 196 channels of data with each channel sampled at rates of 50 samples per second.

A gaseous oxygen system was installed to support the rocket engine operations. It was designed to provide up to 4.0 lbm/sec flow rate at full trailer pressure of 2400 psi. A sonic venturi meter and regulated supply pressure are used to govern the flow rate to the rocket. The same approach is used in the rocket hydrogen system. Automated open loop control of valve position is used to safely bring the rocket to the desired chamber pressure and mixture ratio. Finally, a high pressure water system was also installed to supply water to the rocket coolant channels previously described. A positive displacement pump is used to provide 150 gallons per minute of water at about 1500 psi.

ROCKET RISK MITIGATION TESTS

Rocket mitigation tests were completed on the rocket at the Research Combustion Laboratory (RCL), in Cell 32 prior to installation and tests at ECRL.[7] The purpose of the testing was to quantify rocket performance and develop operating procedures for the direct-connect experiments. Two water-cooled combustion chamber assemblies were tested. Testing was also conducted on a copper heat-sink spool-piece with a water-cooled nozzle to validate injector and ignitor operations.

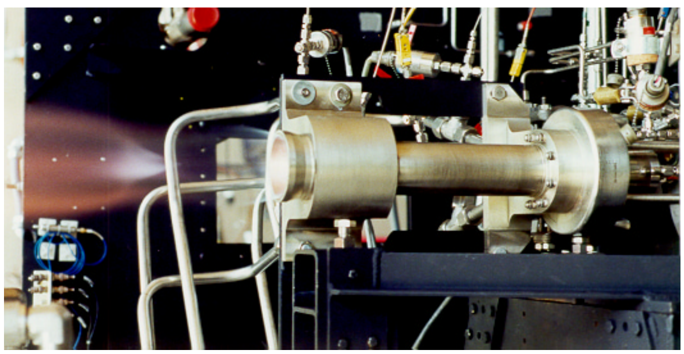


Figure 9 - Rocket chamber during hot-fire testing.

Testing was conducted at chamber pressures of 300 to 750-psia and oxidizer to fuel mixture ratios of 4 and 6. A hot fire is shown in figure 9. Critical performance data generated included characteristic exhaust velocity (C*), specific impulse, and total heat load. For the water-cooled combustion chamber configuration, experimental C* values ranged from 6852 to 7703 ft/sec. Experimental vacuum specific impulse values were in the range of 383 to 424 seconds. Measured values of thrust, combustion chamber pressure, and propellant flow rates were used to determine these performance parameters; measured thrust is converted to vacuum thrust by incorporating ambient pressure. Uncertainty analysis still needs to be completed to assess measurement error (precision and bias) on the reported values. Total heat load was calculated based on water temperature rise through the cooling jacket. Maximum water flow from the facility supply was 16-lb_m/sec. This value was 20% lower than the design value due to higher than expected pressure drop through the coolant channels. Temperature rise was measured between 25 to 57 °F, resulting in 337 BTU/sec to 928 BTU/sec heat loads. Comparisons to performance predictions were made using the TDK computer code.[8] Further information on these results can be found in reference 7.

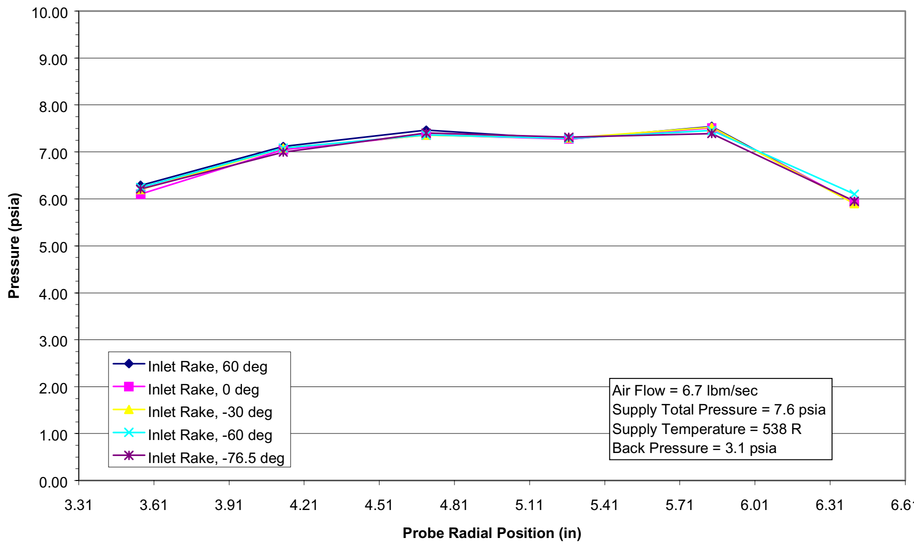


Figure 10 - Combustor entrance pressure profile with center-body fully open.

COMBUSTOR ENTRANCE AIRFLOW CHARACTERIZATION

Unfueled airflow tests were conducted to determine the flow uniformity at the combustor entrance. For these tests, five rakes were installed at the combustor entrance. They were mounted at different circumferential positions, and each contained six equally spaced pitot pressure probes. Two test series were completed to determine radial and circumferential flow uniformity for two different center-body positions.

The first test series was conducted with the center-body located in the full forward position, which provides a nearly constant flow area from the center-body step to combustor entrance. This position represents the initial location of the center-body for lift-off and low speed. The airflow rate and model back pressure were set to correspond to the conditions planned for the initial hot fire tests. As revealed by figure 10, the flow is fairly uniform in both the circumference and radial direction, varying only 1.2 psi from edge to core flow. The circumferential position of each rake is shown in the legend of the figure with the zero degree location at the flow path's centerline. The hub wall is located at 3.31-in. radius and the cowl wall's radius is 6.61 inches.

The second test series was conducted with the center-body located in the Mach 3 flight speed position. In this position, the flow path has a defined converging-diverging area profile with an expansion ratio of approximately 1.95. The tests included both low and elevated model back pressures. The elevated pressures simulated the combustion process. Figure 11 displays the pressure data for the low back pressure case. It reveals greater radial flow distortion than obtained with the fully opened center-body. Flow separation appears to occur at the cowl surface. With elevated back pressures, the flow became more uniform as indicated by mass flow predictions near the cowl wall, but it may not eliminate the flow separation. This pressure profile is displayed in figure 12. Numerical analysis is currently underway to investigate this issue further. It should also be noted that GTX free-jet engine tests reported in reference 9 identified similar concerns, and recommended a reduced inlet turning angle. However, this information was obtained after fabrication of the direct-connect model so it could not be incorporated. It was incorporated into other GTX models. Consequently, further evaluation of the direct-connect experimental data along with ongoing numerical analysis will be used to identify potential modifications to the hardware, if necessary, to improve flow uniformity prior to future direct-connect tests.

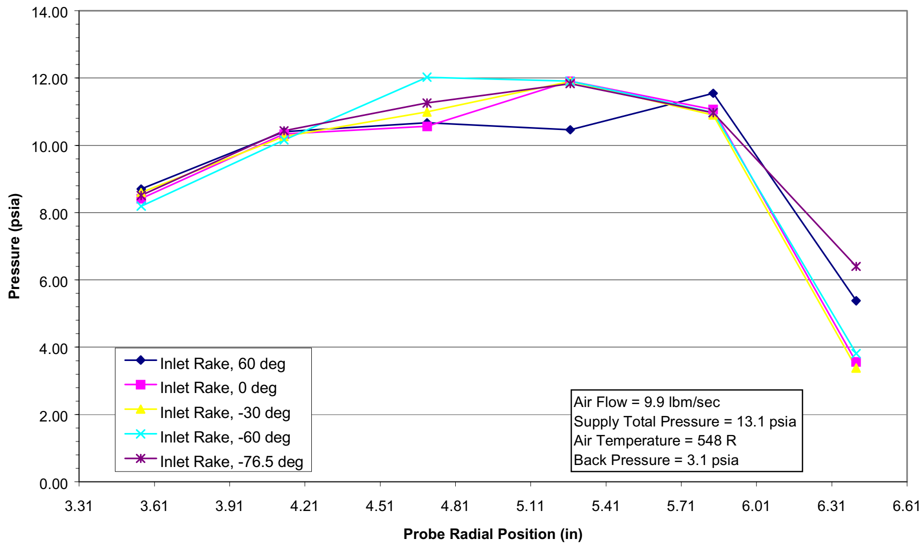


Figure 11 - Combustor entrance pressure profile
With center-body in Mach 3 position, low back pressure.

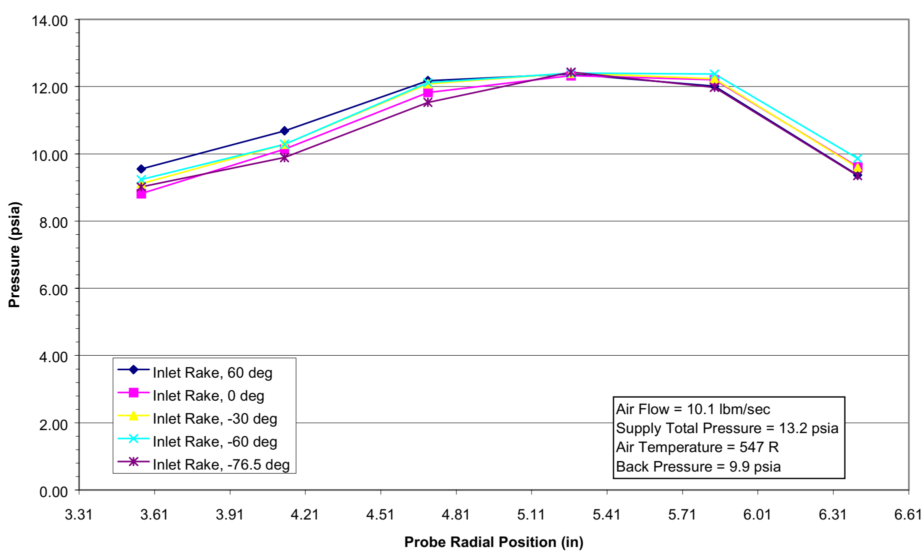


Figure 12 - Combustor entrance pressure profile
With center-body in Mach 3 position, raised back pressure.

INTEGRATED SYSTEM TEST

Finally, a hot fire integrated system test was successfully completed. The tests demonstrated operations of the rocket element, the high pressure cooling water system, the hydrogen system, and oxygen system. This included validating the facility operating procedures and control systems. The checkout tests consisted of operating the rocket element at a combustion chamber pressure of 300 psi and mixture ratios of 4 and 6, for 2 and 5 second durations. The tests were conducted with nominally 3 psia model back pressure, and model airflow was provided to dilute rocket exhaust products. Airflow was 10 lbm/sec at a nominal temperature of 70 °F. The high-speed data systems and gas collection process were also made fully operational.

CONCLUDING REMARKS

An effort was initiated to experimentally investigate low speed operations and performance of a rocket based combined cycle propulsion system. The effort has resulted in the design and fabrication of a direct-connect mixer combustor test article and major facility additions to the Engine Component Research Laboratory. Preliminary tests were completed. These tests included rocket risk mitigation tests, unfueled mixer combustor entrance flow characterization tests, and hot fire integrated system tests. In addition, a set of ejector ramjet tests consisting of 20 hot fires was recently completed. The rocket was operated up to 750 psia chamber pressure and mixture ratios of both 4 and 6. This data is presently being evaluated, and results are forthcoming. Further tests, however, are required to thoroughly investigate low speed operations of an RBCC system. Specific test objectives include comparing the performance of simultaneous mixing and combustion versus independent ramjet stream, and investigating thermal throat, flame propagation, flame stability, and mode transition challenges.

REFERENCES

1. Trefny, C., An Air-Breathing Launch Vehicle Concept for Single-Stage-to-Orbit. AIAA–99–2730, June 1999.
2. Billig, F.S., The Integration of the Rocket with the Ram-Scramjet as a Viable Transatmospheric Accelerator. AIAA–93–7017, 11th International Symposium on Air-Breathing Engines, Sept. 1993.
3. Yungster, S., Trefny, C.J., Analysis of a New Rocket-Based Combined Cycle Engine Concept at Low Speed. AIAA–99–2393, June 1999.
4. Steffen, C.J., Yungster, S., Computational Analysis of the Combustion Process in an Axisymmetric, RBCC Flowpath. JANNAF 2000 APS–CS–1135, November 12–17, 2000.
5. Repas, G.A., Hydrogen-Oxygen Torch Ignitor. NASA TM–106493, March 1994.
6. Colantonio, R.O., Pachlhofer, P.M., Parrott, E.L., Piendl, B.R., Advanced Combustor Rig at NASA Lewis Research Center's Engine Components Research Laboratory. NASA TM–4546, March 1994.
7. Smith, T.D., Walker, J.F., Rig 1 Rocket Test Report—Phase 1. GTX Technical Note 01–003.
8. Nickerson, G.R., Berker, D.R., Coats, D.E., Dunn, S.S., Two-Dimensional Kinetics (TDK) Nozzle Performance Computer Program, Volume III, Users Manual. NAS8–39048, March 1993.
9. Thomas, S.R., Saunders, J.D., Walker, J.F., Mach 3.5 to 7 Free-Jet Engine Tests of the GTX RBCC Flowpath With a 12 Degree-Spike Inlet. NASA/TM—2000-210528, Dec. 2000.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)

2. REPORT DATE
May 2002

3. REPORT TYPE AND DATES COVERED
Technical Memorandum

4. TITLE AND SUBTITLE
Status of the RBCC Direct-Connect Mixer Combustor Experiment

5. FUNDING NUMBERS
WU–708–90–63–00

6. AUTHOR(S)
James F. Walker, Hani Kamhawi, Thomas M. Krivanek,
Scott R. Thomas, and Timothy D. Smith

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
John H. Glenn Research Center at Lewis Field
Cleveland, Ohio 44135–3191

8. PERFORMING ORGANIZATION REPORT NUMBER
E–13334

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546–0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA TM—2002-211555

11. SUPPLEMENTARY NOTES
Prepared for the Combustion, Airbreathing Propulsion, Propulsion Systems Hazards, and Modelling and Simulation Subcommittees Joint Meeting sponsored by the Joint Army-Navy-NASA-Air Force, Destin, Florida, April 8–12, 2002. Responsible person, James F. Walker, organization code 5880, 216–977–7465.

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Categories: 07 and 20 Distribution: Nonstandard

Available electronically at http://gltrs.grc.nasa.gov/GLTRS
This publication is available from the NASA Center for AeroSpace Information, 301–621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***
The NASA Glenn Research Center is developing hydrogen based combined cycle propulsion technology for a single-stage-to-orbit launch vehicle application under a project called GTX. Rocket Based Combined Cycle (RBCC) propulsion systems incorporate one or more rocket engines into an airbreathing flow path to increase specific impulse as compared to an all rocket-powered vehicle. In support of this effort, an RBCC direct-connect test capability was established at the Engine Components Research Laboratory to investigate low speed, ejector ramjet, and initial ramjet operations and performance. The facility and test article enables the evaluation of two candidate low speed operating schemes; the simultaneous mixing and combustion (SMC) and independent ramjet stream (IRS). The SMC operating scheme is based on the fuel rich operations of the rocket where performance depends upon mixing between the rocket plume and airstream. In contrast, the IRS scheme fuels the airstream separately and uses the rocket plume to ignite the fuel-air mixture. This paper describes the test hardware and facility upgrades installed to support the RBCC tests. It also defines and discusses low speed technical challenges being addressed by the experiments. Finally, preliminary test results, including rocket risk mitigating tests, unfueled airflow tests, and the integrated system hot fire test will be presented.

14. SUBJECT TERMS
Combined cycle propulsion; Rocket-based; Combustion; Experiment

15. NUMBER OF PAGES
16

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	